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EVALUATION OF THE PILOT ROCKET SCRUBBER.(U)

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EVALUATION OF THE PILOT ROCKET SCRUBBER

FINAL REPORT

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JANUARY 1978

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DIRECTOR OF SCIENCE AND TECHNOLOGY
AIR FORCE SYSTEMS COMMAND
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NOTICES

FOREWORD

This report covers work on Project 573012YT (formerly 573012CH and 683MEEYX) "Rocket Exhaust Scrubber Technology", by the Test and Support Division of the Air Force Rocket Propulsion Laboratory (from September 1970 to January 1978).

The project engineers were Milburn Raleigh for the design and construction phase, and Jack E. Hewes and Luciano Sedillo for the buildup and testing phase.

This report has been reviewed by the Information Office/XOJ and is releasable to the National Technical Information Service (NTIS). At NTIS it will be available to the general public, including foreign nations.

This report is unclassified and suitable for public release.

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20. ABSTRACT (Continue on reverse side if necessary and identify by block number)

The technical and economical feasibility of a pilot rocket scrubber was evaluated based on 23 tests of liquid and solid rocket motors with a thrust of 5,000 pounds. The initial test, using liquid fluorine and gaseous hydrogen, resulted in extensive damage to the test system. After switching to storable liquid propellants, 18 tests were conducted to characterize the scrubber system. The main concern of the Air Force was the results from scrubbing four solid rocket motors. This report summarizes the history, and describes the over

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installation, testing, evaluation, and costs for operating the pilot rocket scrubber.

While the pilot scrubber appeared to cleanse the exhaust products, HCl and Al_2O_3, only very basic solid rocket conditions were addressed. It was determined that the pressure and temperature ratings of the demister were very inadequate. The most important discovery of this study was the presence of afterburning of exhaust products, CO and H_2. The costs involved in installation, operation, maintenance, and modification are discussed. It was recommended the rocket motor scrubbers not be used because of the many technical problems and large program costs. For the pilot rocket scrubber, the total program cost was found to be about four times the original installation cost.

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I. INTRODUCTION

In FY 70, the Air Force Rocket Propulsion Laboratory was concerned about environmental pollution and decided to develop the technology to reduce the level of exhaust emissions from experimental rocket testing to acceptable standards. This study was to determine the technical and economical feasibility of a family of scrubbers in the event they became mandatory for rocket testing.

The AFRPL and the Arnold Engineering Development Center (AEDC) had a common interest, since both orgainzations test rockets at ambient and altitude simulated conditions. The AFRPL requested AEDC to do a study of the state-of-the-art of gas cleaning for liquid and solid motors and to provide a design for a pilot rocket scrubber. The AFRPL project engineer, Mr. M. Raleigh, provided design criteria and technical guidance for this effort. ARO, Inc. (subsidiary of Sverdup & Parcel and Associates) under contract F40600-73-C-0004 to AEDC started its design work in September 1970 and completed it in August 1972. Their work is documented in Reference 1, a literature and equipment survey, resulting in the selection of a high gas velocity chemical spray scrubber. This study is applicable to scrubbers for use with rockets from 1,000 to 250,000 pounds of thrust. It included a point design for a subscale "pilot exhaust scrubber" to accommodate a 5,000 poind thrust rocket motor. This scrubber was fabricated, installed, and tested by the AFRPL. The operating instructions for the scrubber are presented in Reference 2. This system was to be capable of removing HF, HCl and Al_2O_3.

The AFRPL also contracted with the Environmental Protection Agency (EPA) to "define the technology required for reducing the gaseous and particulate emissions from rocket motor firings." The EPA then contracted with Air Pollution Technology, Inc., (APT) on EPA contracts 68-02-1328 and 68-02-2145. APT's initial efforts resulted in Reference 3 which evaluated present technology for treating exhausts. Reference 4 is their report providing design criteria and preliminary cost of large rocket exhaust scrubbers for use with rockets with up to 450,000 lbs of thrust.

This report summarizes the history, and describes the installation, testing, evaluation, and costs for operating the 5,000 lb thrust rocket exhaust scrubber. It also relates the operating experience and costs involved to establish the practicality of using scrubbers on rocket test stands.

II. SCRUBBER EVALUATION

APPROACH

The main concern of the AFRPL was the scrubbing of solid rocket motors. However, because of the operational characteristics of solid motors such as start-up transients, irreversible starts, etc., the system was activated using a proven liquid rocket engine. Liquid rocket engines are easier to scrub because the mass flow rate, oxidizer to fuel ratio, and chamber pressure are controllable. Another important feature of the liquid rocket engine is that it may be stopped at will in the event of a malfunction.

Once the scrubber was well characterized, the solid rocket motors were to be tested. An end burning solid propellant grain configuration was selected, because its start transient was very gradual compared to center perforated grains. The initial solid propellant grains were to have a burn time of 10 seconds; the final test grains were to burn for 30 seconds.

Various parameters throughout the system were to be measured such as: temperatures, pressures, and gas compositions. Gas samples were to be analyzed by the AFRPL, and particulate samples were analyzed by APT. It was decided not to measure thrust because of the additional expense to design and fabricate a thrust stand. While it is possible to calculate thrust from our measurements, it is not possible to determine the effect of the scrubber on thrust measurements.

APT was to provide particulate samplers and a design for momentum reducers. The introduction of momentum reducers was the result of APT's initial study (Reference 3). Their purpose was to attempt to minimize the amount of

scrubber solution necessary to neutralize the HCl and the gas velocity. The solution was to be collected by momentum reducers that redirected the water from along the duct walls 90° back into the exhaust gas stream. The AFRPL was to install and test the particulate samplers and to fabricate, install, and test the momentum reducers. The counting and sizing of the particulates was to be done by APT.

The test plan for evaluating the scrubber was initially as follows:

Test No. 1	10 second checkout grain
Test No. 2	30 second grain
Test No. 3	30 second grain with scoop momentum reducers installed
Test No. 4	Repeat of Test No. 3
Test No. 5	30 second grain with plate momentum reducers installed
Test No. 6	Repeat of Test No. 5
Test No. 7	30 second grain with most effective momentum reducer
Test No. 8	Repeat of Test No. 7

TEST SYSTEM

The entire test system shown in Figure 1 consisted of a 5,000 lb thrust solid rocket, a pilot scrubber, two solution tanks, and an evaporation pond.

The pilot rocket scrubber was designed by ARO, Inc., and reported by Garrett, el al (Reference 1). The scrubber, presented in Figures 1 and 2, consisted of the following components: a diffuser, a spray section, a mixing section, and a demister (entrained moisture eliminator). This particular design was selected for the following reasons: "(1) the high rocket gas velocity can be used to achieve high efficiency in cleaning solids and gases, (2) changing spray solutions allows a particular contaminating gas to be removed, (3) the spraying solution cools the gases, and (4) the spray systems and ducting are simple and economical to construct." AEDC documented the pilot scrubber and its installation in detail in their drawings RWT01503-26 sheets.

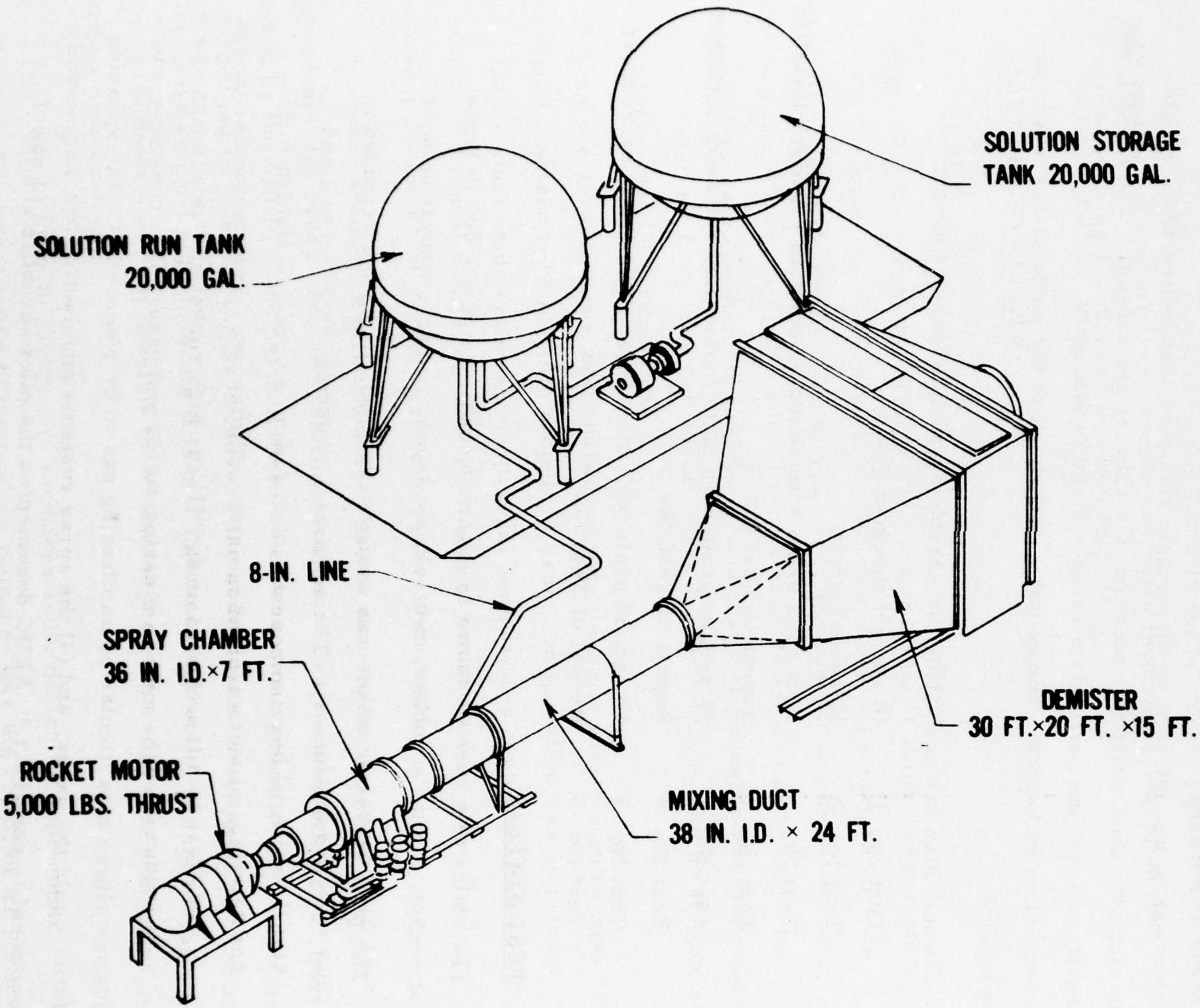

Figure 1. Pilot Rocket Scrubber

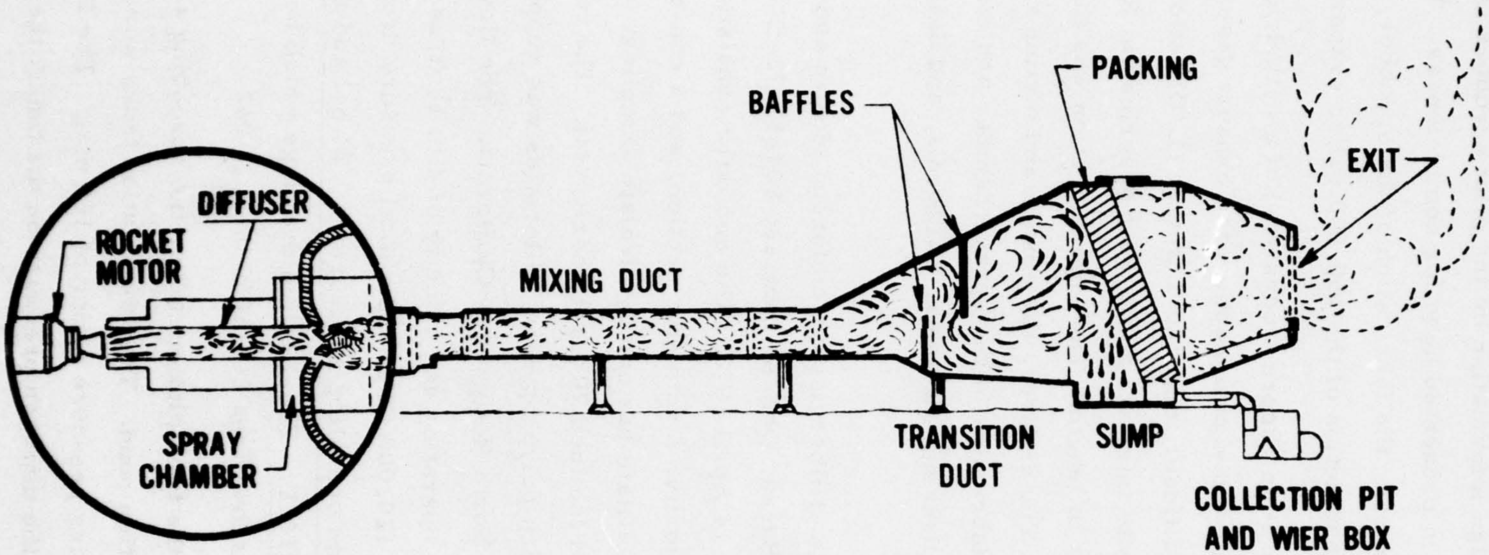

Figure 2. Cross Section of Pilot Rocket Scrubber

Although the detail design information on the pilot scrubber is presented in Reference 1, a description is presented here for completeness. The test system was acquired o.. three contracts: the fabrication of the scrubber, the fabrication of the demister, and the construction of the 150,000 gallon evaporation pond.

The spray chamber shown in Figure 3 consisted of a cylindrical diffuser with a converging conical inlet design to capture the exhaust gases. The diffuser had a diameter of 8" and a diffuser length to diameter (L/D) ratio of 8. The APT injector design was quite intricate and is best described in Reference 1. The spray chamber was 7 ft in length. The mixing section was three feet in diameter and 24 feet long. The diffuser, the spray, and mixing sections were constructed of steel. The fabricator of the spray chamber and mixing duct was constructed by Chalco Engineering, Gardena, California, and delivered to AFRPL on 18 May 1972.

The demister was made of fiberglass, reinforced plastic and manufactured by the Celicote Company, Berea, Ohio on contract F04611-72-C-0029. It was delivered to the AFRPL on 14 April 1972. The demister consisted of diverging ducting to lower the gas velocity, a demister section, and a converging conical 7 foot diameter outlet to facilitate gas and particulate sampling. The demisting section was rectangular and inclined 70^{o} from horizontal. The two foot packing section wall filled with 1 inch polypropylene Tellerettes was supplied by the John M. Vossler and Co., North Hollywood, California. The flow area was 240 ft^2 and provided a safe operating gas velocity of 4 to 10 ft/sec. The demister was designed for 120,000 CFM and a total pressure drop of 3 inches of water. It had a maximum operating pressure of 0.21 psi and a maximum operating temperature of 210^{o}F. The detailed drawings which best describe the demister are Celicote's drawings D-HA039 through 042.

Figures 4 and 5 show the two solution tanks, the scrubbing section, the demister, and the evaporation pond. The two solution tanks were 20,000 gallon capacity and had an operating pressure rating of 165 psig. The flow of solution was controlled by varying the nitrogen pressure on the tanks; the relation

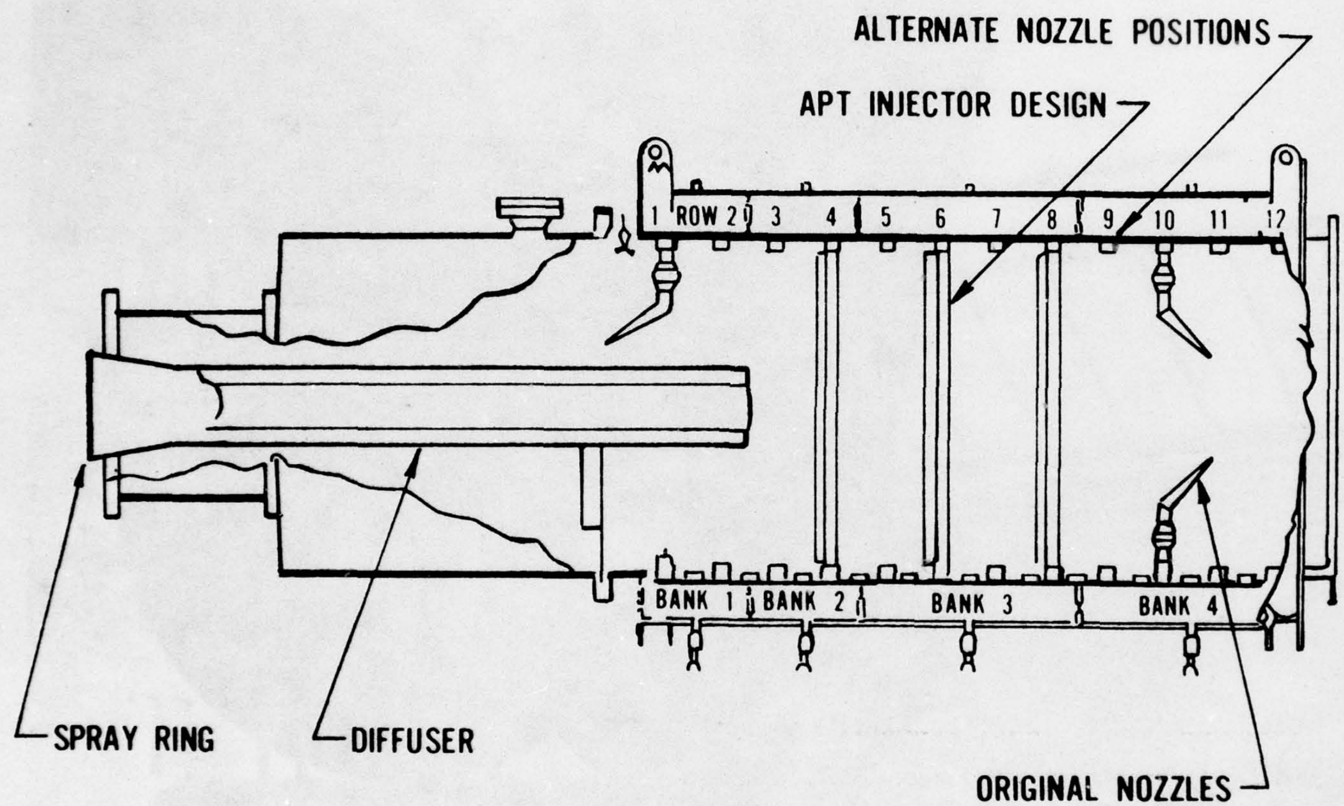

Figure 3. Inlet Details of Scrubber

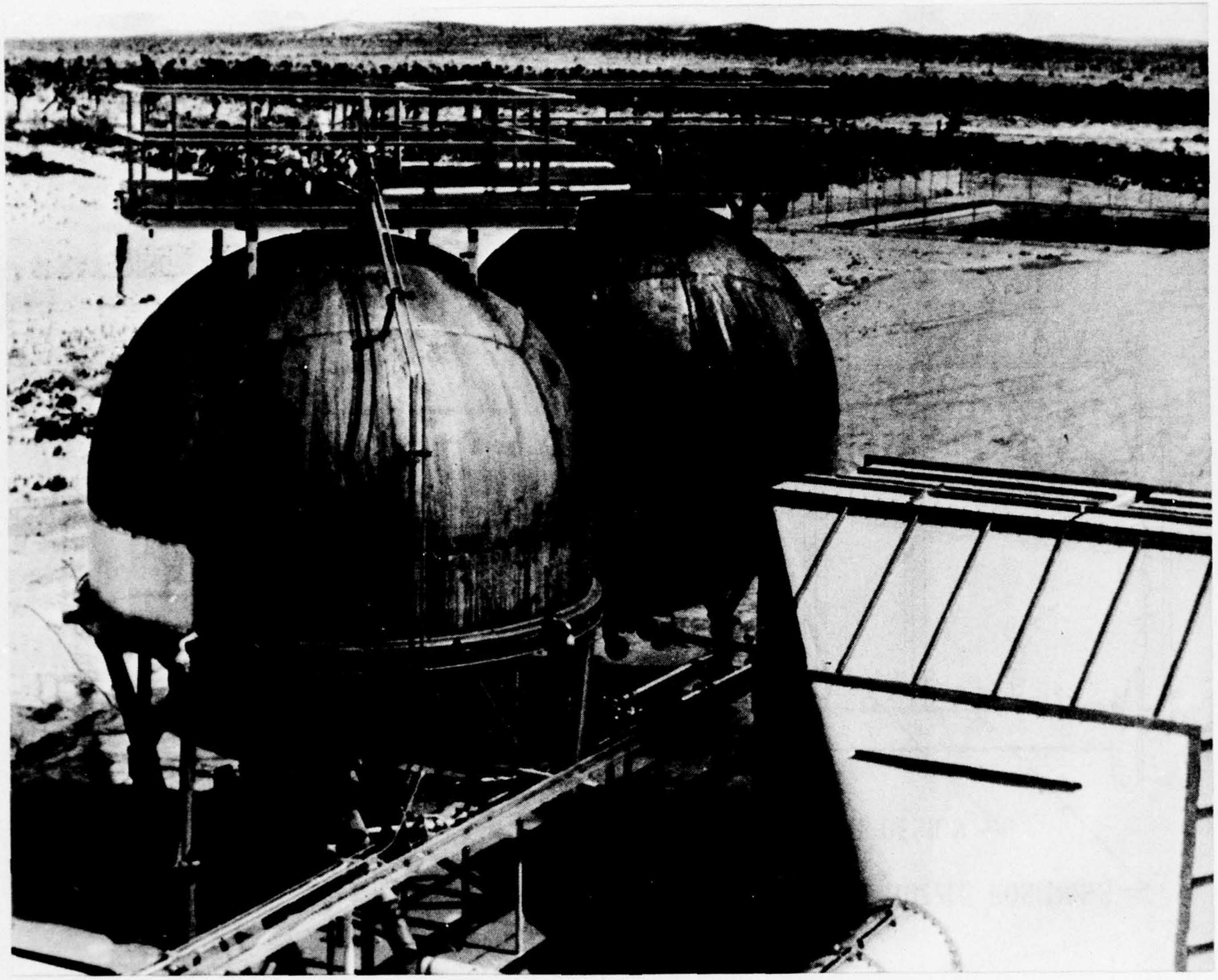

Figure 4. Top View of Scrubbing System

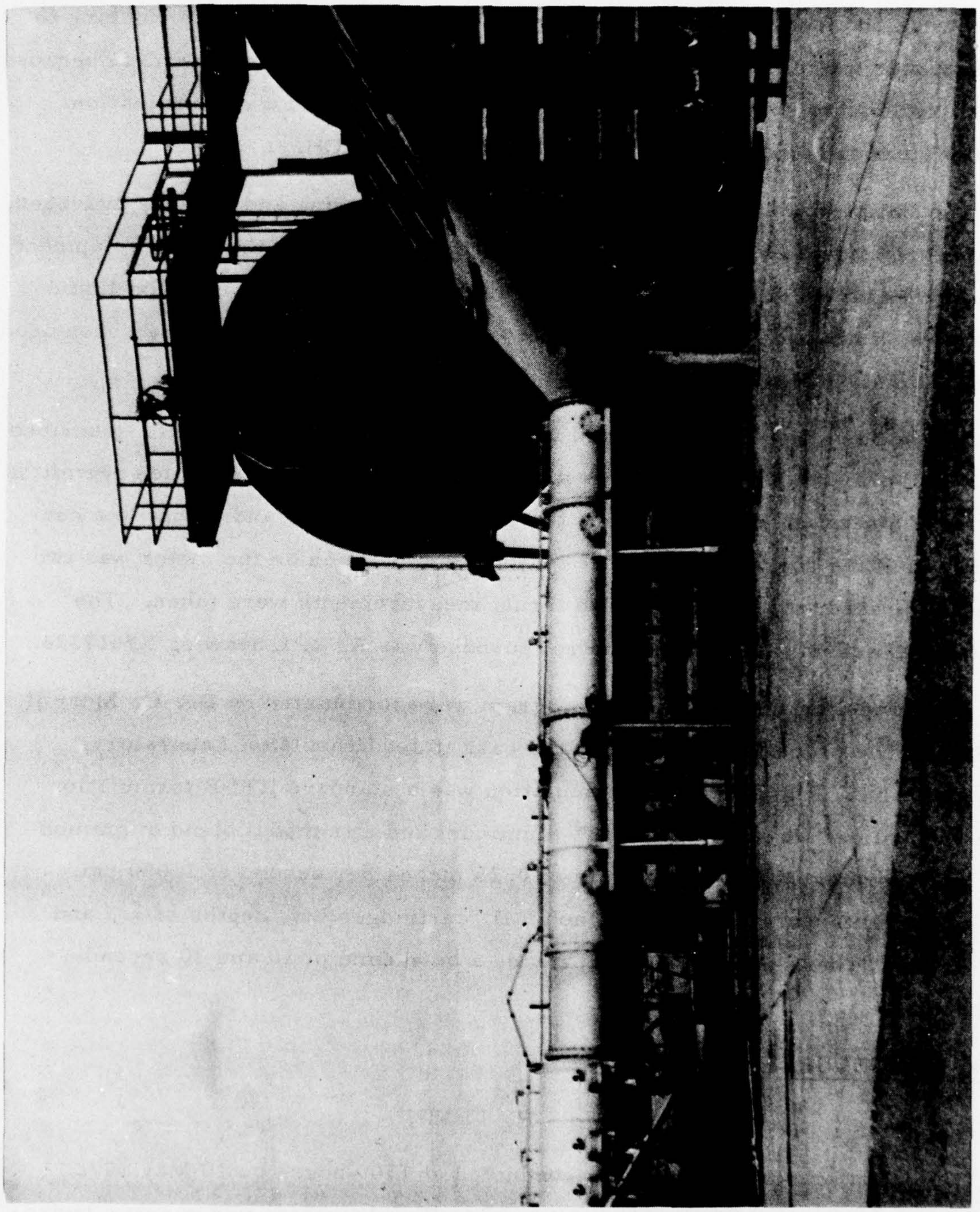

Figure 5. Side View of Scrubbing System

between solution flow rate and tank pressure is presented in Figure 6. These tanks were obtained from the AFRPL inventory. The line from the run tank to the scrubber was 8 inches. KOH was selected as the scrubbing chemical because of its excellent solubility in water, and it presented less of a crystallization problem than the other candidates, i. e., CaOH and NaOH.

The initial liquid rocket engines, using liquid fluorine and gaseous hydrogen, were proven liquid engines that had been used on the Center Engine Development Program and the Thrust Chamber Thermal Fatigue Program. The next liquid engine was a nitrogen tetraoxide and Aerozine-50 liquid engine that had been used on the Flintstone Program.

The solid rocket motor selected for these tests was the 40" CHAR described in AFRPL drawing X7317220. This motor was a heavy duty type which permitted cartridge loaded end-burning grains to be used. Figures 7 and 8 show the configuration of the rocket motor. The only instrumentation on the motor was two standard pressure transducers. No thrust measurements were taken. The mechanical schematic used for control purpose was AFRPL drawing X7517324.

The propellant grains for this program were formulated by Dr. C. Merrill, AFRPL. The propellant was mixed and cast at Jet Propulsion Laboratory, Edwards AFB. The propellant formulation was a standard HTPB formulation with 88% solids. It consisted of 16% aluminum and a trimodal blend of ammonium perchlorate. It had a burn rate of 0. 25 inches per second at 1, 000 psi. The propellants were cast into 36 inch I. D. cartridges with depths of 3. 5 and 9. 5 inches. These grains were to provide a burn time of 10 and 30 seconds, respectively.

TEST PROCEDURE

F_2 - H_2 TEST

The initial checkout firing was conducted at 1700 hours on 10 May 1972. A proven in-house liquid rocket engine, using liquid fluorine and gaseous hydrogen, was programmed for a steady state combustor pressure of 500 milliseconds

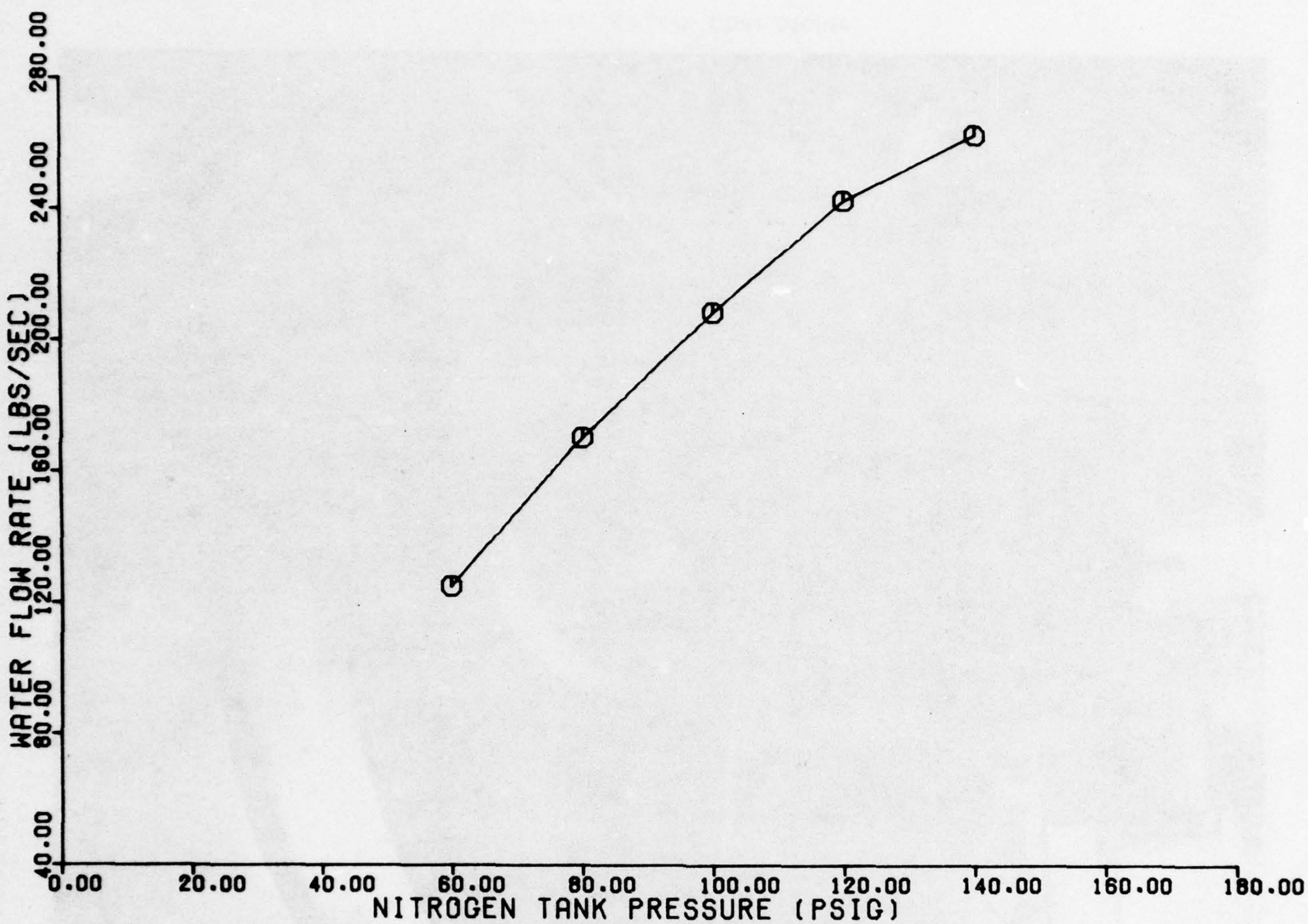

Figure 6. KOH Solution Flow Rate

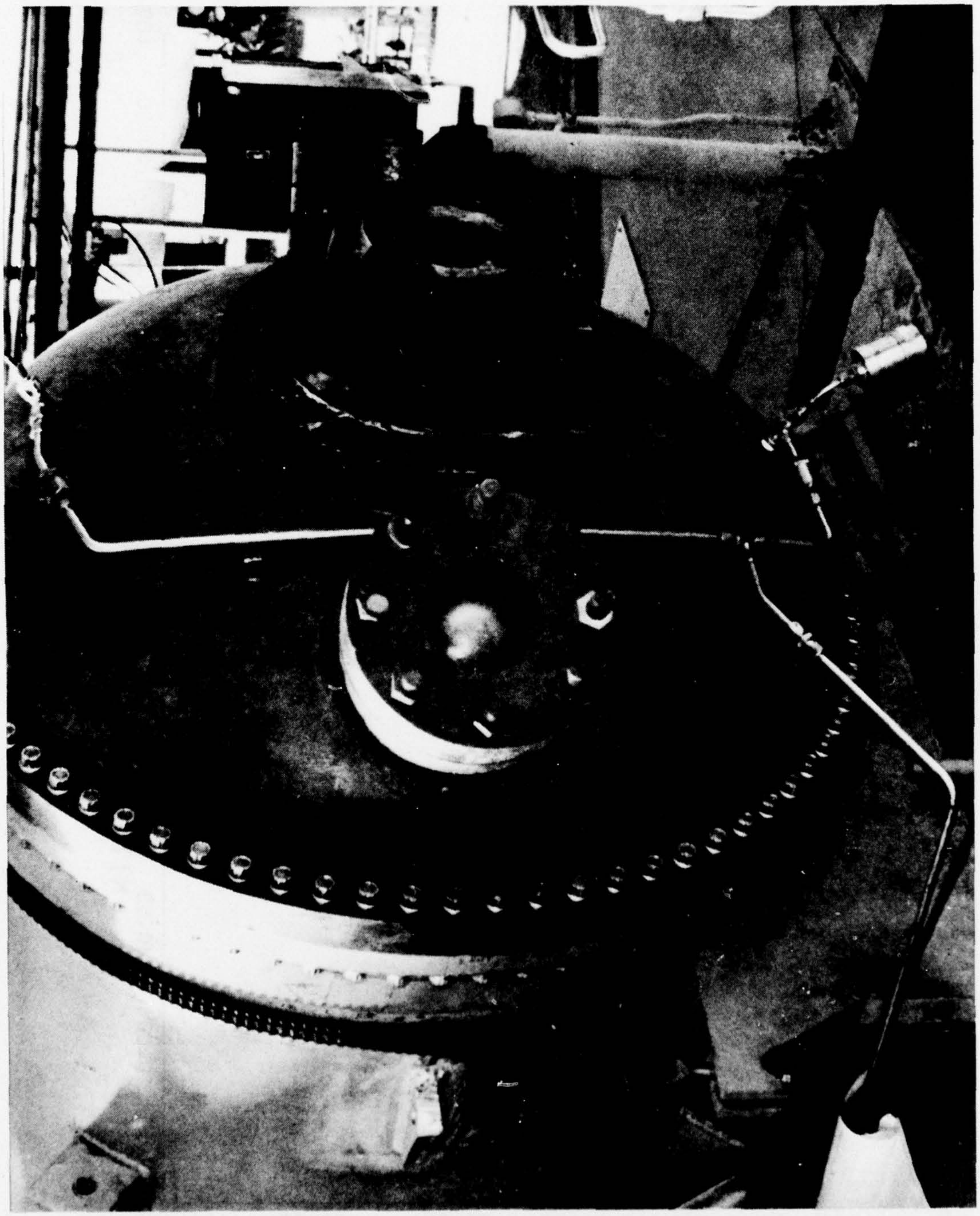

Figure 7. 40 Inch Char Motor

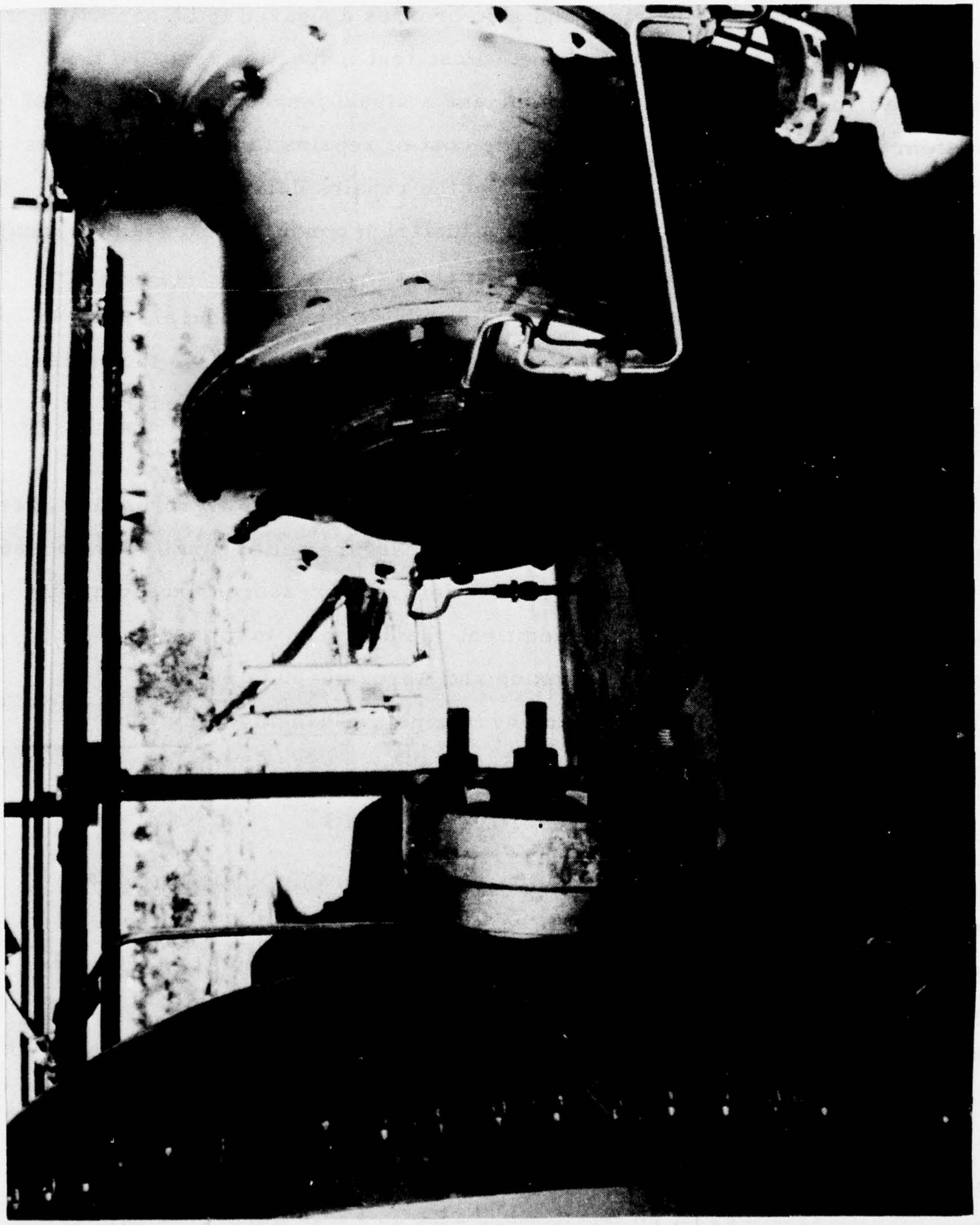

Figure 8. 40 Inch Char Motor-Scrubber Interface

duration. The test was terminated after 200 milliseconds by a low chamber pressure sensor. The engine start and stop profiles appeared to be normal from previous operating history of this engine. Post-test inspection revealed the demister had sustained extensive damage, and a visual inspection of the rest of the system revealed no other damage. The cost of repairs to the demister was $3,000 for materials and 600 manhours, and the repairs delayed this project 6 months. The problem was identified as a fuel-rich condition (excess hydrogen) which was ignited by electrostatic charges. The pressure spike was observed about two seconds after engine cutoff on the scrubber duct transducers, and it varied from 0.145 psig upstream to 4.6 psig downstream just above the entrance of the fiberglass demister module. The pressure spike occurred nearly simultaneously at both ends of the duct.

The conclusion was that the demister failed from internal overpressurization. To prevent reoccurrence of this kind, 18"x24" pressure relief panels were added which were closed by gravity but provided relief from pressures exceeding the pressure rating of the demister. In addition, the decision was made to switch to storable propellants, nitrogen tetroxide and Aerozine-50 (a 50-50 mixture of unsymmetrical dimethyhydrazine and hydrazine). Changing to storable propellants reduced the possibility of having explosive gases in the demister.

N_2O_4/AEROZINE - 50 TESTS

While the demister was being rebuilt, the new storable liquid propellant system was installed. The mechanical schematic for the N_2O_4/Aerozine-50 system is documented on AFRPL drawing X7317215. The initial test with this system occurred at 1445 hours on 13 December 1973, and there were 18 total tests with this system through 8 January 1975. There were some tests required for the purpose of "softening" engine start transients and to assure that relief panels operated properly. Several problems occurred which delayed the test program. The engine required various modifications while working out the starting transients and had a 6" ID chamber with an L* of 100". The oxidizer to fuel ratio was 2.0, the nominal flowrates were 5 lbs/sec of fuel and 10 lbs/sec

of oxidizer, and the chamber pressure was 750 psig. The scrubber KOH solution was 1% KOH and had a flowrate of 1000 gpm. The test duration was gradually increased to 7.5 sec, and gas samples were taken during the last three tests with no success.

These tests were used to work out modifications and improvements on the scrubber system before attempting to test solid rocket motors. Modifications were made to the demister by adding pressure relief panels, removing some internal baffles, and increasing the instrumentation in the demister.

COMPOSITE SOLID PROPELLANT TESTS

Between January 1975 and June 1975, the facility A Pad in Test Area 1-52 was modified to accept a solid rocket motor having a thrust of 5,000 lbs.

III. DISCUSSION OF RESULTS

ROCKET MOTOR PERFORMANCE

The motor selected was the 40" CHAR which is a heavy duty boiler plate type reusable solid rocket motor. This motor has thick insulation throughout and can be reused with a minimum of expense, rework, and time delay. It is shown in Figures 7 and 8. It was decided not to take thrust measurements because of the lack of a suitable thrust stand. The only instrumentation on the rocket motor was two standard transducers for measuring chamber pressure. Four tests were done during this phase of the program as presented below:

TABLE 1

DATES OF SOLID ROCKET TESTS

Test Number 1	8 October 1975	10 second duration
Test Number 2	27 October 1976	10 second duration
Test Number 3	5 November 1976	10 second duration
Test Number 4	19 November 1976	10 second duration

It is important to note that all tests were 10 seconds in duration; the program did not progress beyond the checkout phase for reasons that will be addressed later. None of the 30 second duration grains were tested. The solid rocket motor characteristics are shown in Table 2.

TABLE 2

SOLID ROCKET MOTOR CHARACTERISTICS

Nozzle Diameter	2 inches
Thrust	5,000 lbf
Specific Impulse	262 s
Mass Flow Rate	18.9 lbm/s
Volume Flow Rate, actual	95,000 ACFM
Temperature, static	3,780^{o}R

Since a typical solid motor start transient would severely damage the demister, the ignition characteristics of the 40" CHAR motor were tailored for a smooth gradual ignition (see Figure 9). A small test series was conducted at Test Area 1-32 to develop and verify the ignition characteristics. A typical motor pressure versus time trace is shown in Figure 9.

The five pressure relief panels, 18"x24", did not permit pressures in excess of the demister rating. However, much of the exhaust product escaped before reaching the demister. These panels would open an average of five times during a 10 second test. An interesting observation was that steam was very visible in the exhaust through the pressure panel but not in the exhaust from the demister exit. The exhaust from the demister exit was clear with heat waves clearly visible.

HEAT DAMAGE

The gas and particulate samplers located at the demister exit were extensively damaged by heat during the first test. This event was unexpected, since the sampling system had easily survived the liquid engine tests. On the second

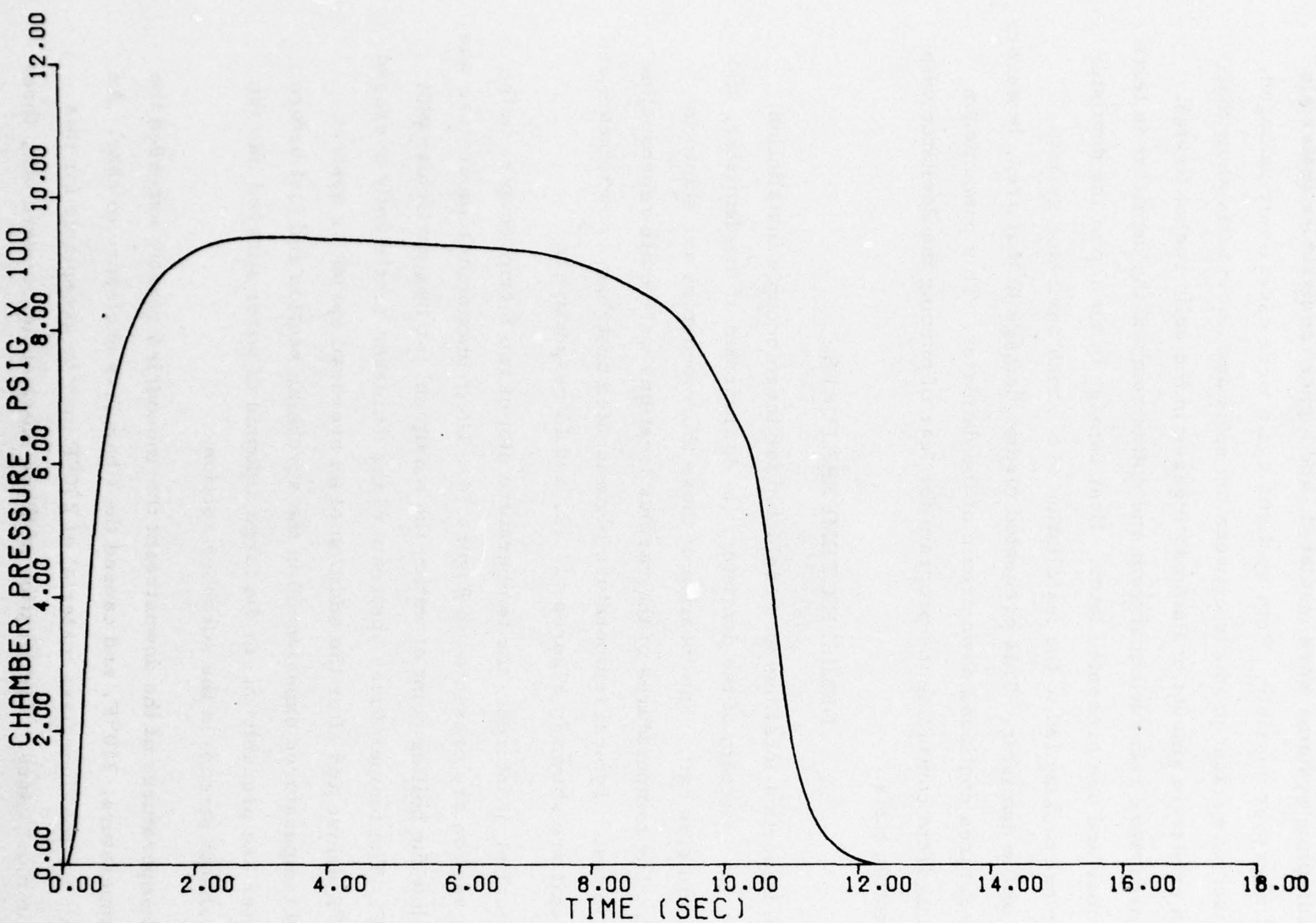

Figure 9. Chamber Pressure

test, the sampling systems were insulated, and higher temperature materials were used wherever possible. Both systems again were extensively damaged. A comparison of exhaust products between the solid and liquid tests show that there is an excessive amount of flammable gases in the solid rocket firings. Temperatures were recorded upstream and downstream of the demister in tests three and four and are presented later. Heat damage to the top of the demister and uppermost packing led to the installation of a 2 inch sprinkler system upstream of the demister. This prevented further damage to that area, however, high temperatures continued downstream of the demister. This phenomenon prevented us from continuing the program for fear of burning the demister with longer duration tests.

DEMISTER TEMPERATURES

Three important locations were selected for thermocouple installation; they were: (a) upstream of the demister, (b) downstream of the demister, and (c) at the demister exit. The location of these thermocouples are shown in Figure 10. The temperatures of the various locations were quite reproducible from test to test. Typical temperature plots located upstream, downstream,and demister exit are shown in Figures 11, 12, and 13 respectively.

As a matter of interest, the temperature plot of two thermocouples in the scrubbing section are presented in Figure 14. Their maximum temperature was always below the boiling point of water; the maximum for this particular plot was 185°F. The temperatures upstream of the demister consistently averaged about 180°F before and after the addition of an overhead sprinkler system. These low temperatures experienced in the scrubbing section and just before the demister are probably due to the large amounts of water sprayed into the gaseous exhaust stream in the scrubber section.

The temperatures of the downstream thermocouples greatly exceeded the design temperature, 210°F, and caused the fiberglass enclosure to char. An upper limit for the fiberglass material of 250°F may be acceptable for this material in most cases. The location of these thermocouples were one, three

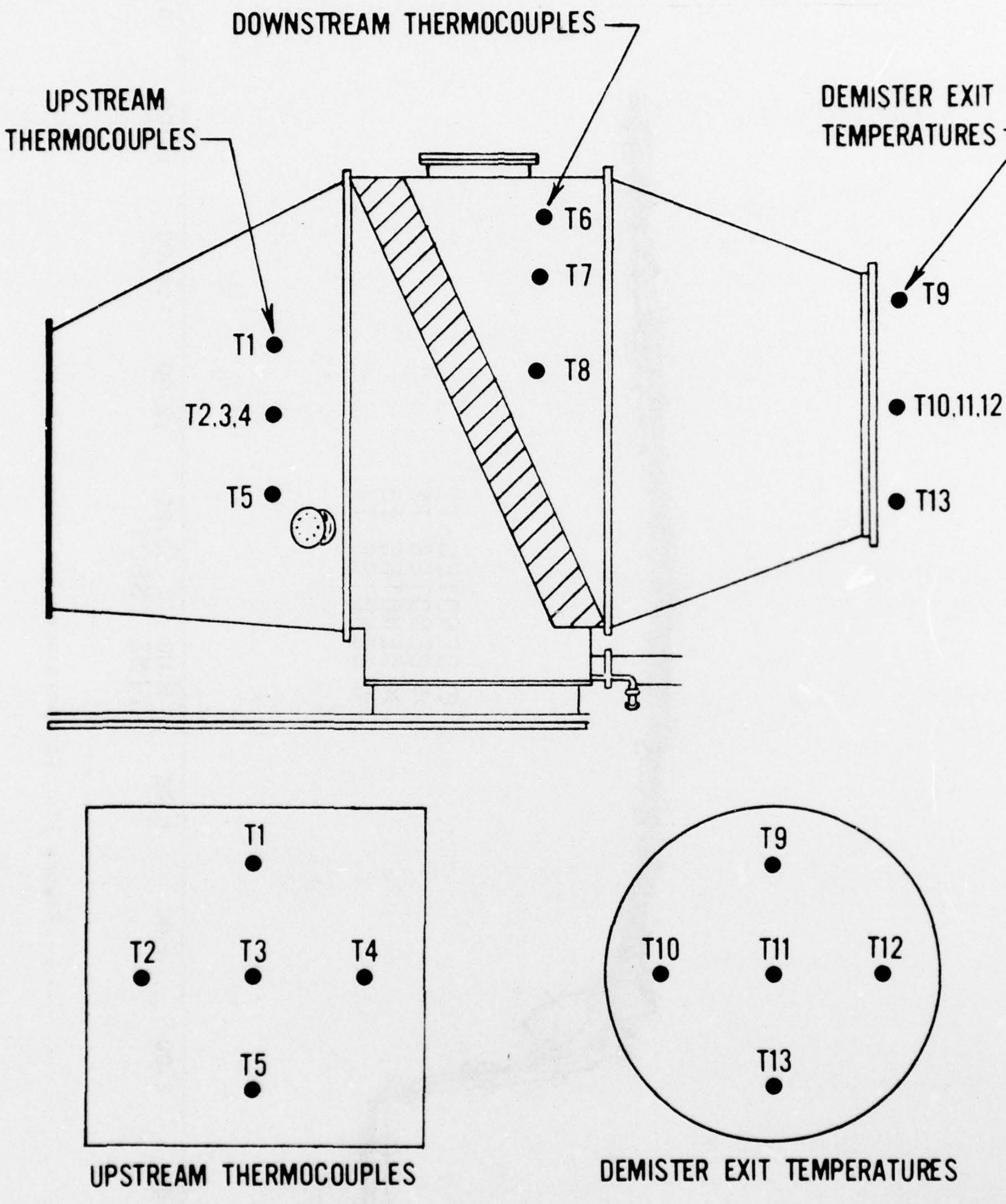

Figure 10. Demister Thermocouple Location

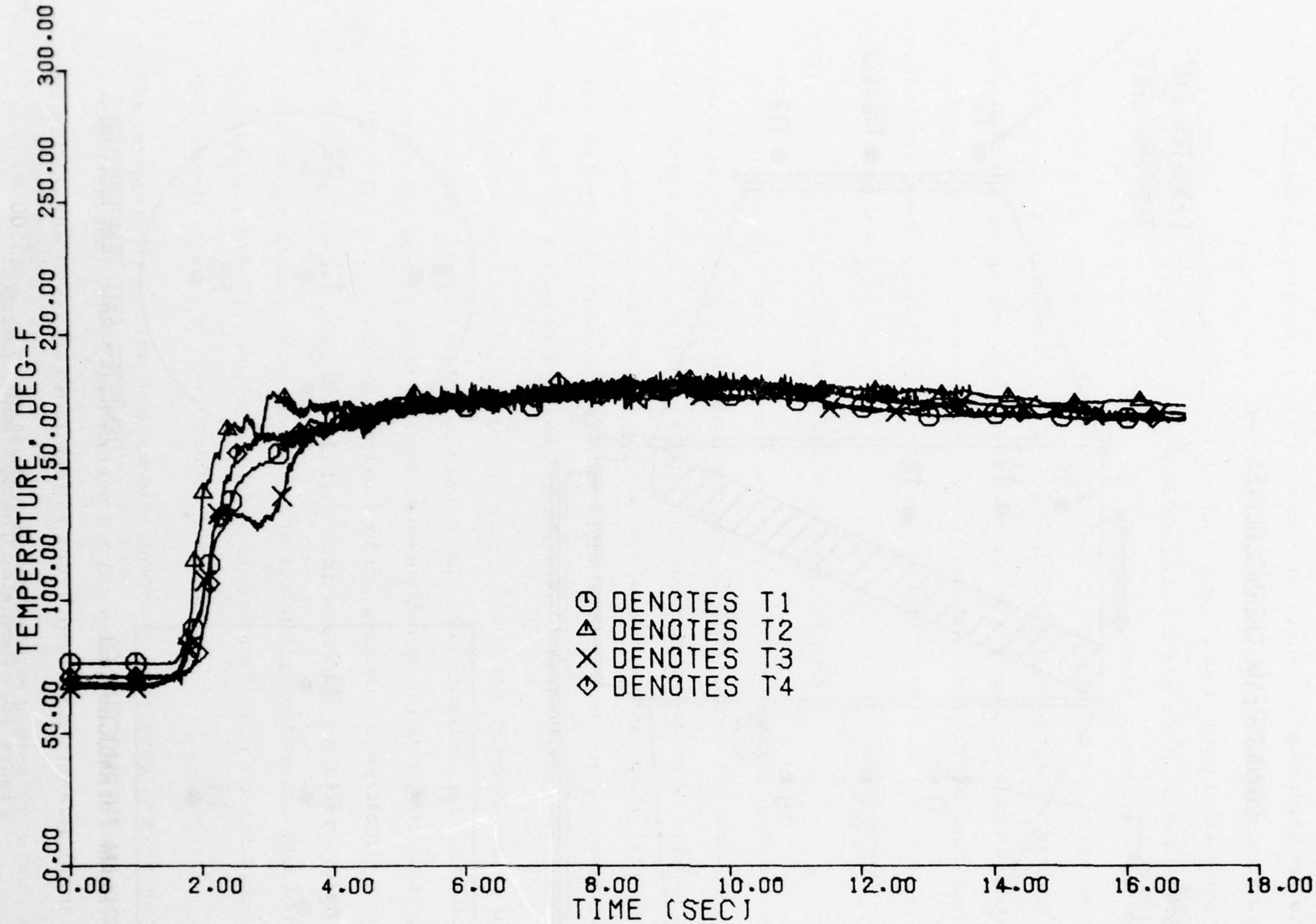

Figure 11. Temperatures Upstream of Packing

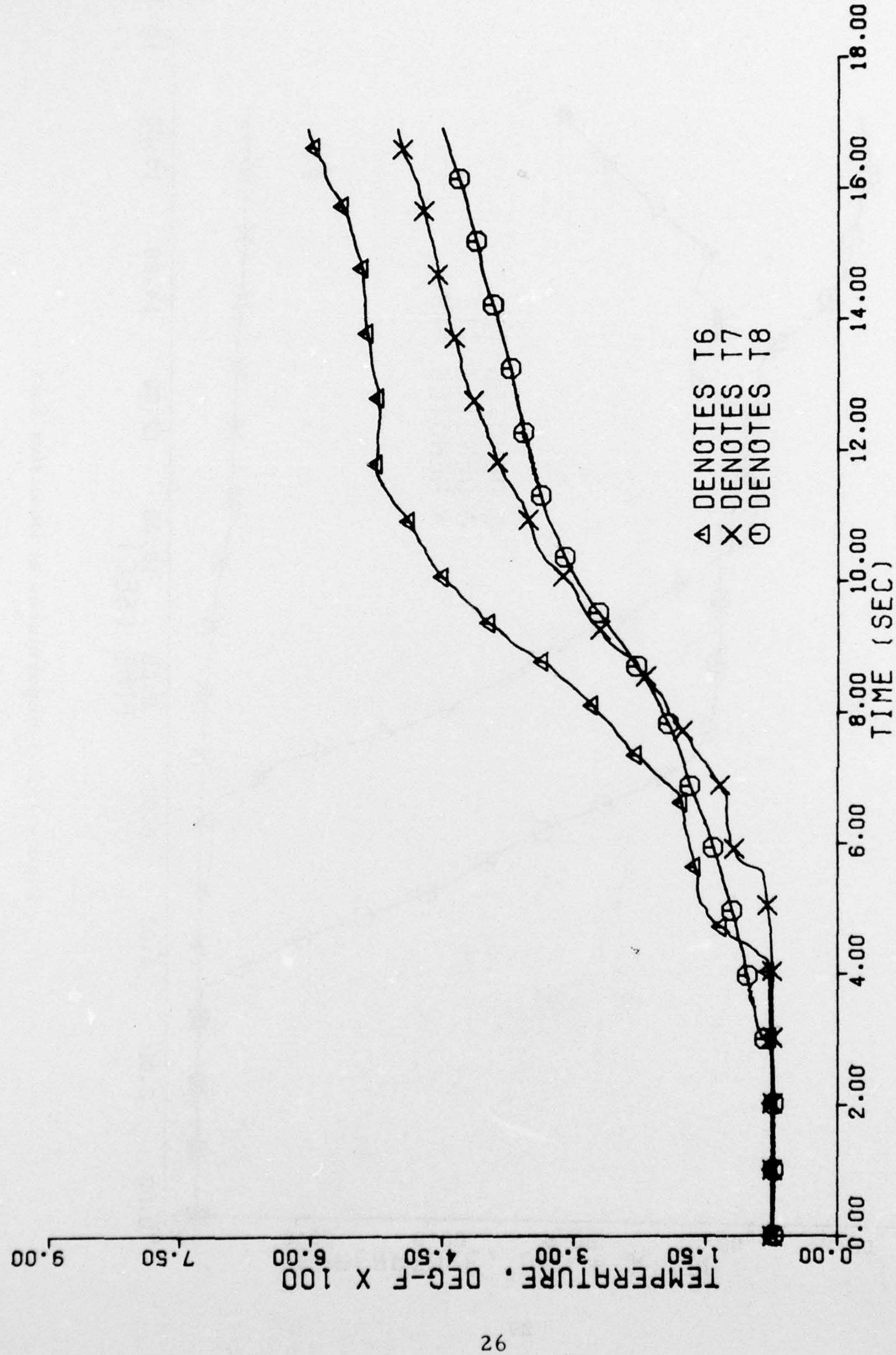

Figure 12. Temperatures Downstream of Packing

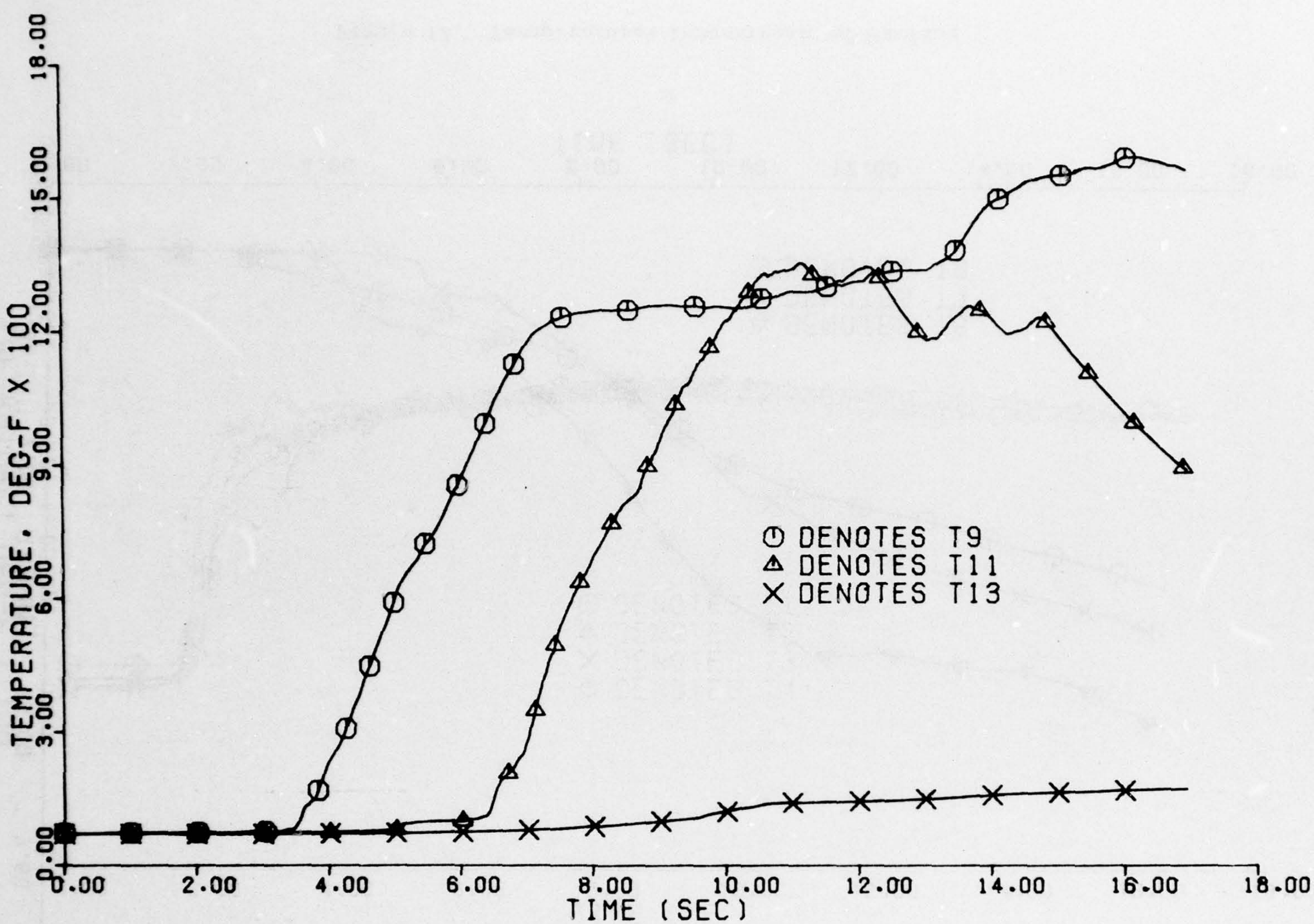

Figure 13. Temperatures at Demister Exit

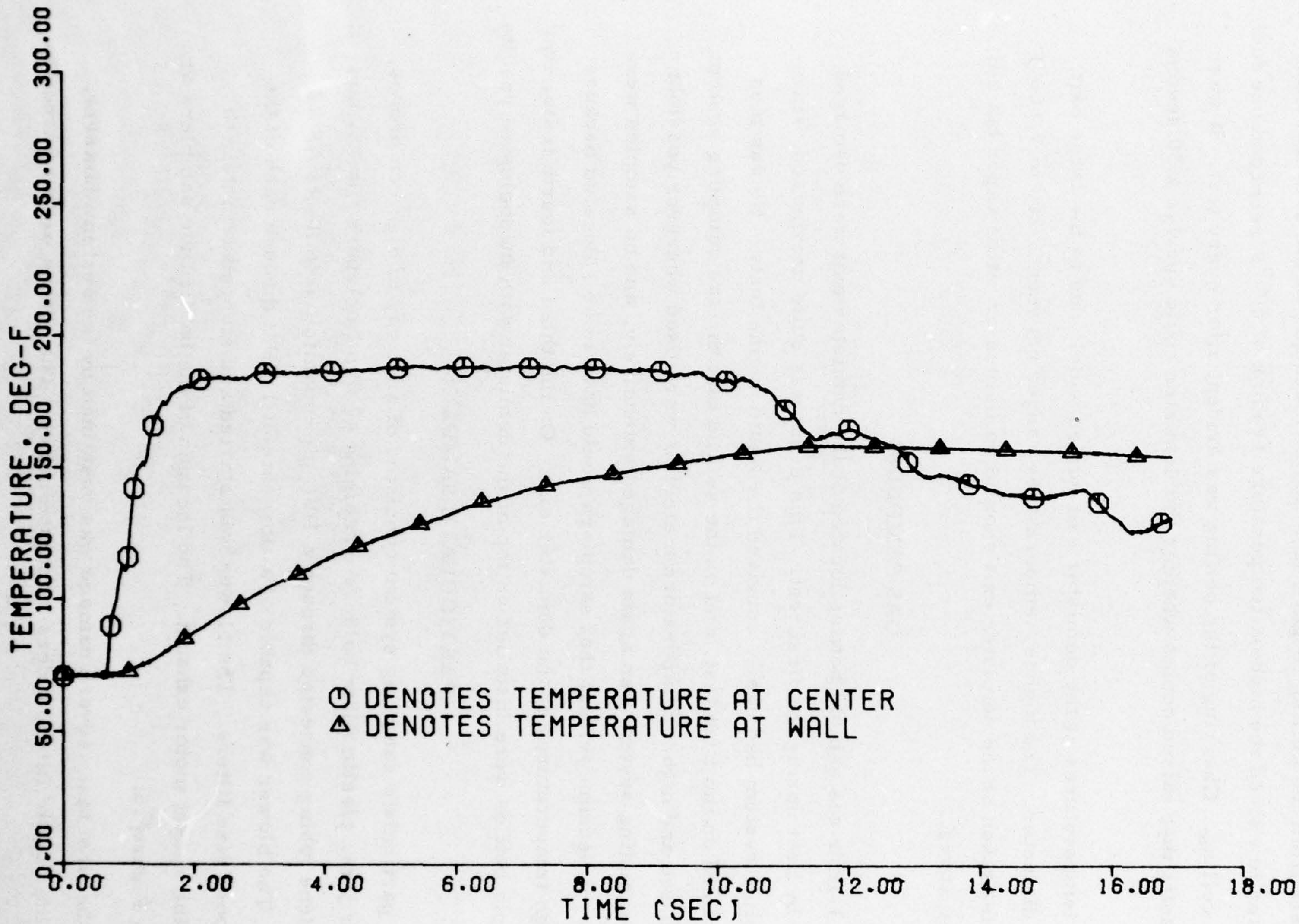

Figure 14. Temperatures Inside Scrubbing Section

and six feet from the ceiling (Figure 10). A thermocouple at the ceiling would be expected to exceed the highest temperature reading of 600°F recorded one foot from the ceiling. Charring of the ceiling was evident after every test. It was these results that raised doubts whether the demister could survive a 30 second test.

The temperatures at the demister exit varied widely due to the large exit, 7 feet in diameter. The highest temperatures ranged between 1,200 to 1,600°F. The movies taken of the demister exit showed no flames or water vapor but did show heat waves.

GAS SAMPLING

The 1-liter gas sample bottles located at the demister exit were damaged severely by heat during the first test. This event was quite unexpected, since the sampling system had easily survived the liquid engine tests. No samples were obtained on the first test, and on the second test the gas sampling system was insulated and higher temperature materials were used wherever possible. The gas sampling system was again damaged extensively, and no samples were obtained. It became evident that samplers would have to be relocated because of the high temperatures at the demister exit. On the third and fourth tests, two gas sample bottles were installed on top of the demister with inconclusive results.

PARTICULATE SAMPLING

The particulate sampling system consisted of an array of eighteen probes, precutter jars, plastic filter holders containing 47 mm Nuclepore filters, and polyethylene tubing connected through a PVC pipe manifold to a Roots AF 22 blower. The blower was capable of a flow rate of 1 CFM through each of the 0.8 um pore size filters. The blower was started and stopped remotely to capture the rocket motor exhaust. The location of the particulate samplers are shown in Figure 15.

On the first test, severe damage was sustained by the system; however, most of the plastic filter holders remained intact. The filters were returned to

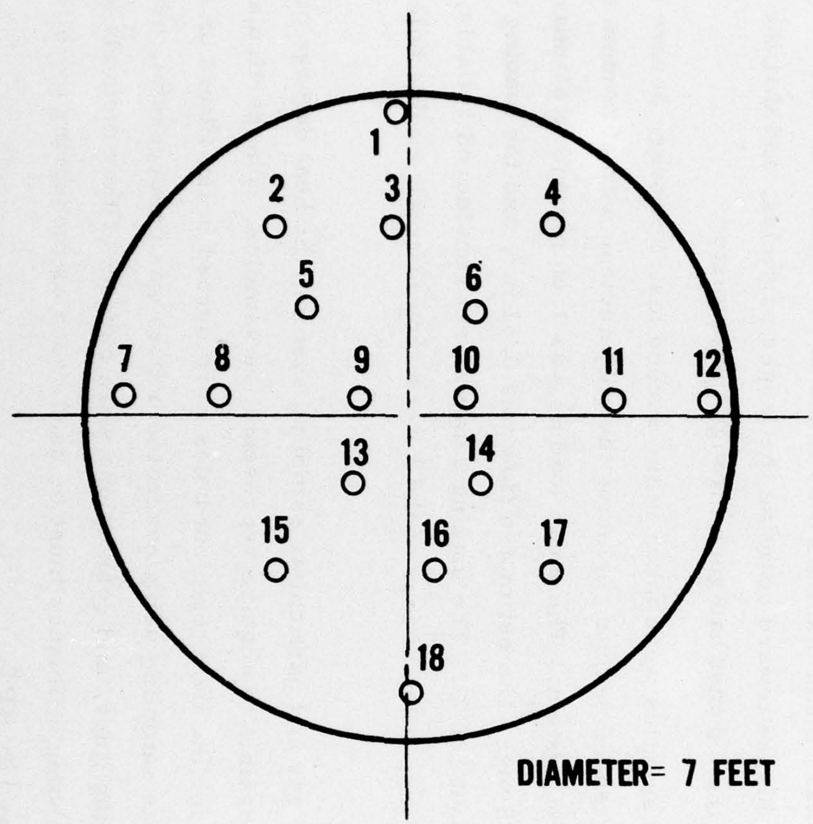

Figure 15. Location of Particulate Samplers

APT for gravimetric and optical sizing analysis. The mean weight gain was 0.8 milligrams (mg), and the standard deviation was 0.3 mg. Since the pore size of the filters was 0.8 μm, particles larger than 0.8 μm diameter are retained 100%, and smaller particles retention was less efficient. If burning of the tubing occurred after the sampling period, the mean particulate loading would have been 5 mg/m^3. This represents a particulate collection efficiency of 99.9%. A detailed analysis of the particle count and analysis is presented in Reference 4. The geometric mass mean diameter was found to be 9 μm. APT cautions that the technique is biased toward counting the larger particles, and that the sample was taken after the exhaust had gone through the demister.

Particulate samples were taken in the second test; however, severe damage was again experienced despite replacing the polyethylene with aluminum tubing and insulating the system. The mean loading was 1.66 mg with a standard deviation of 0.78 mg. The estimated flow was 0.51 m^3, and the loading was 33 mg/m^3 at about 300^{o}C. The loading data was not considered reliable, because post test visual inspection of four of the filters showed small visible holes due to melting.

On both the gas and particulate sampling system, the heat damage made it practically impossible to acquire representative samples. The particulate system worked for the first test, but holes were burned in the filters on the second test. Gas sampling using evacuated bottles was unsuccessful, because holders, sampling lines, and connections were burned. Other methods of measuring scrubbing efficiency must be used, such as measuring the effluent liquid as proposed by APT.

NEUTRALIZING AND COOLING SOLUTION

The solution used to neutralize HCl contained 3% by weight KOH and had a flow rate of 240 lbs/sec. The solid propellant grain weighted nominal 200 lbs and was consumed in 10 seconds to yield a propellant mass flow of 20 lbs/sec. Therefore, the ratio of KOH solution to propellant was 12:1. This amount of solution was adequate, however it may be more than required. From Reference 4,

it appears that a cooling solution could be plain water, and the resulting acidic effluent neutralized in the evaporation pond.

AFTERBURNING

The first indication of exhaust gas afterburning was the progressive temperature increase through the demister. The temperatures at the demister exit ranging between 1200°F and 1600°F indicated that CO and H_2 reacted with oxygen. The various mechanisms which could have produced these temperatures were: heat of mixing HCl with water, reaction of KOH with aluminum, and burning of flammable gases in the exhaust. The only mechanism that could produce these high temperatures is the burning of flammable gases. The most probable source of ignition is electrostatic charges which buildup when fluids flow through ducts. From an environmental pollution standpoint, burning CO and H_2 is desirable, but it would be better to burn it outside the demister. Several methods have been proposed to accomplish afterburning without damage to the demister: (a) remove the exit section of the demister whose primary purpose was to facilitate sampling, (b) discharge the exhaust in a vertical direction, and (c) restrict the air supply.

An interesting observation is the difference between the gas composition at the nozzle exit and downstream of the demister. The scrubbing section lowers the gas stream temperature to about 180°F and absorbs all the HCl into the water. The demister removes essentially all entrained water droplets and aluminum particulates. The composition of the exhaust gas products at the nozzle exit and downstream of the demister are shown in Table 3. The gas immediately downstream of the demister is assumed to be at 180°F and saturated with water. The air injected into scrubbing system was ignored. From these considerations it can be seen that the percentage of flammable gases downstream of the demister is 38.8% by volume.

TABLE 3

COMPOSITION OF ROCKET EXHAUST GASES

Component	Molecular Weight	Nozzle Exit Volume %	Downstream of Demister Volume %
H_2O (g)	18	20.3	50.7
HCl	36.5	16.8	-
Al_2O_3 (s)	102	7.0	-
N_2	28	8.5	7.5
CO_2	44	3.6	3.2
CO	28	20.3	17.9
H_2	2	23.5	20.7

IV. COST SUMMARY

This section attempts to establish a realistic basis for estimating total scrubber program costs based on our experience with the 5000 lb thrust pilot scrubber. The limited amount of testing normally associated with any rocket research of development program results in the test system being reconfigured frequently. The operation and maintenance costs far exceed the initial installation costs. This was the case with the pilot scrubber as shown below:

TABLE 4

INITIAL INSTALLATION COSTS

Engineering design	$ 75,000
Earthwork	32,000
Scrubber	58,000
Demister	38,000
Installation	44,000
Instrumentation and Controls	52,000
TOTAL	$299,000

TABLE 5

OPERATION, MAINTENANCE AND MODIFICATION COSTS

Manpower (direct)	$377,000
Operation, Maintenance & Modification	649,000
TOTAL	$1,016,000

The total cost of the program is the sum of initial installation costs and the operational, maintenance and modification costs. For the pilot scrubber, the total program cost was $1,325,000.

The installation costs agree well with the estimates presented by Garrett, et al, (Reference 1) who estimated an installed cost of $252,000 in 1971 compared to the actual cost of $299,000. The cost estimate for this size scrubber is given by Reference 4 as $215,000 in terms of 1976 costs. The difference between these costs is probably the original engineering study that preceded the design work which increased the engineering costs on the AFRPL scrubber. The operation and maintenance costs are overstated by about $200,000 due to difficulties encountered in the test program. For purposes which will become evident later, I propose to use a total program cost of $1,125,000 as a more representative figure. Then the ratio of the test program cost to the initial installation cost is 3.76.

For a rocket with a thrust of 450,000 lbs, APT (Reference 4) estimates $4,480,000 for installation costs of a conventional design and $1,680,000 for a high risk unconventional design. Both of these estimates are based on a horizontal test stand and a minimum amount of engineering to accomplish the overall design. Our experience with our pilot scrubber (assuming no catastrophic occurrence) would indicate that the total program for a 450,000 lb thrust rocket scrubber costs to be:

Total Program Costs for Conventional Scrubber Design

$4,480,000 X 3.76 = $16,845,000

Total Program Costs for an Unconventional Scrubber Design

$4,680,000 X 3.76 = $6,317,000

Our experience at the AFRPL shows that the testing of four large rocket tests in the 450,000 lbs thrust level would be roughly equivalent to conducting 23 tests at 5,000 lbs thrust level.

The point that needs to be made in this section is that the installation costs are only part of the total program cost. For this test program, the installation costs were 23 or 27 per cent of the total program costs depending on the total costs that are used. For purpose of estimation, the total program cost of a rocket scrubber should be estimated at 4 times the initial installation cost.

V. CONCLUSIONS AND RECOMMENDATIONS

The pilot scrubber appeared to do its intended task of cleansing the exhaust products of a rocket motor. Studies by Reference 4 indicated that 99% plus of all particulates are easily removed, and 3 feet of reactive length is all that is required to put gaseous HCl into solution with water. The absence of steam vapor at the demister exit indicated that the demister removed excess moisture. Our attempts to sample the gas stream were ineffectual.

An important discovery was the presence of afterburning which greatly influences the design of future scrubbers. The gases, CO and H_2, are ignited by electrostatic charges and cause extremely high temperatures downstream of the demister. This important phenomenon was overlooked in the original design.

The most important conclusion was that the original design of the demister was very inadequate. The scrubbing and mixing section were adequate except that the design of the water nozzles needed additional attention. The pressure and temperature design limits of the demister were underrated. Pressure transients quite common in rocketry would severely damage the present demister. No recommendations on pressure rating can be given, since pressure measurements couldn't be taken without damaging the demister. Temperatures downstream of the demister greatly exceeded design temperatures. Only the short duration of

the tests prevented damage to the demister; the longer burning grains would surely have burned the demister. It is proposed that the exit section downstream of the demister be removed or a higher temperature material be used.

Another important conclusion was that the two foot section of packing could have been reduced to a one foot section as originally planned; this change would also reduce the pressure drop across the packing. Visual examination of the exhaust indicated that the demister was performing its job of removing entrained moisture.

This program addressed only the preliminary technical feasibility of scrubbing exhaust gas products. No attempt was made to refine the designs or to investigate the impact of scrubber systems on testing experimental rocket systems, such as performance measurements, thrust vector control, testing at altitude, etc.

It is recommended that scrubbers not be installed on rocket test stands, because the addition of a scrubber greatly complicates the operation of testing rockets. Much damage is sustained by the scrubber after each test, and these scrubber tests were done under very controlled conditions using very judicious grain selection to minimize damage from start transients. Normal rocket test operations would greatly increase the damage. Excessive amounts of time and effort were spent correcting the damage or making improvements to prevent further damage. Additionally, in rocket testing, it is not unusual to eject the nozzle, nozzle adapter, and aft closure. Such an event would greatly damage a scrubbing system. Experimental rocket motors have also been known to explode, and this would result in replacing the entire scrubber, an expensive proposition.

While it may be technically feasible to scrub rocket exhausts, it is economically prohibitive. The test program to evaluate the 5,000 lb pilot rocket scrubber cost the Air Force $1,325,000 for 23 tests; the average cost per test was $57,600. The projected cost for a 450,000 lb rocket scrubber suitable for the Super Hippo motor is $16,845,000. These costs do not reflect any damages resulting from catastrophic events. Another item that is commonly discussed

is the "installation cost" of a scrubber. It turns out that the installation cost is about 25% of the total cost of a test program.

REFERENCES

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